NASA TECHNICAL
MEMORANDUM

NASA TM X-53075

AUGUST 12, 1963

FACILITY FORM 602

N64-32486
(ACCESSION NUMBER)

(THRU)

29
(PAGES)

1
(CODE)

TMX-53075
(NASA CR OR TMX OR AD NUMBER)

31
(CATEGORY)

NASA TM X-53075

PROPULSION DEVELOPMENT PROBLEMS ASSOCIATED WITH LARGE LIQUID ROCKETS

by A. A. McCOOL AND G. H. McKAY, JR.
Propulsion and Vehicle Engineering Laboratory

OTS PRICE

XEROX	$ 2.00 FS
MICROFILM	$ .50 MF

NASA

George C. Marshall Space Flight Center, Huntsville, Alabama

TECHNICAL MEMORANDUM X-53075

PROPULSION DEVELOPMENT PROBLEMS ASSOCIATED WITH LARGE LIQUID ROCKETS

by

A. A. McCool

G. H. McKay, Jr.

George C. Marshall Space Flight Center
Huntsville, Alabama

ABSTRACT

32486

Propulsion development problems of the SATURN I and SATURN V vehicles are reviewed. These two vehicles are typical of large multi-stage vehicles containing many interfaces, producing a multitude of integration problems that must be solved during development. Of the many problems encountered in the SATURN I program, only the most significant ones are presented. The SATURN V program has not progressed to the test phase; therefore, only design problems are considered.

Author

NASA-GEORGE C. MARSHALL SPACE FLIGHT CENTER

NASA-GEORGE C. MARSHALL SPACE FLIGHT CENTER

TECHNICAL MEMORANDUM X-53075

PROPULSION DEVELOPMENT PROBLEMS ASSOCIATED WITH LARGE LIQUID ROCKETS

by

A. A. McCool

G. H. McKay, Jr.

PROPULSION AND MECHANICS BRANCH
PROPULSION AND VEHICLE ENGINEERING DIVISION
RESEARCH AND DEVELOPMENT OPERATIONS

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

TECHNICAL MEMORANDUM X-53075

PROPULSION DEVELOPMENT PROBLEMS ASSOCIATED WITH LARGE LIQUID ROCKETS

by

A. A. McCool

G. H. McKay, Jr.

SUMMARY

This treatise, presented to the First Annual Propulsion Meeting of the American Institute of Aeronautics and Astronautics*, reviews propulsion development problems of the SATURN I and SATURN V vehicles. These are typical of large multistage vehicles containing many interfaces, producing many integration problems to be solved during development.

The significant problem areas are described with the philosophy that led to workable solutions and the developments that produced a near-to-ideal design in the SATURN stages. Only the more significant problems in the SATURN I program are presented; since SATURN V has not reached the test phase, only design problems are considered.

INTRODUCTION

Many of the propulsion problems that must be solved during development of large multistage space vehicles are created by design complexity. Design decisions are sometimes influenced by expediency, such as the cluster concept of the SATURN I utilizing existing tooling for tankage and development of the H-1 engine based on the engines of the JUPITER and ATLAS. Adapting these readily available components to a new vehicle configuration produces additional design and development problems. Creation of an all new and much larger space vehicle like the SATURN V poses additional new problems based solely on size. Added to these are the multitude of technological problems associated with clustered large cryogenic propellant engines.

* This treatise was not published by the American Institute of Aeronautics and Astronautics.

SECTION I. CLUSTER TANKAGE PROBLEMS

The SATURN I first stage employs a cluster concept utilizing nine propellant tanks and eight engines. Although the clustering concept is not the most sophisticated design possible, the decision to use this arrangement was predicated on the ready availability of the necessary tooling for the tankage and also on the early availability of a simplified engine system based on the JUPITER and ATLAS engine design. It was clear from the beginning that many design and development problems would be associated with this configuration.

Following the initial designs, a quarter-scale model was constructed for evaluation of fluid flow characteristics. As may be seen from FIG 1, LOX in the center tank is not directly fed to the engine cluster, but is transferred to the four outboard LOX tanks, each of which supplies two engines. Scale model flow tests showed that the upward flow of the transferred propellant seriously interfered with the downward flow into the suction lines feeding the engines. This condition resulted in a premature gas breakthrough into the suction lines that, if uncorrected, would result in large residuals in the tanks at burnout. These residuals were estimated at about 16, 000 pounds. Effective solution was to separate the two opposing flow patterns by some distance. Restrictions imposed by the tank diameter prohibited this separation in a horizontal plane. Therefore, it was provided in the vertical direction by adding a 30 inch standpipe to the end of the transfer line. Installation of the standpipe necessitated pressurizing the center tank approximately 6 psi higher than the outboard tanks to overcome the additional gravity head introduced. Although considerable turbulence was observed in the liquid during the terminal draining, it appeared that the problem had been solved.

Orifice sizes, necessary to achieve 6 psi pressure differential between the center tank and the outboard tanks, were calculated and installed for one of the early static tests. During LOX tanking, it was discovered that the orifices were too small to permit adequate venting of the outboard LOX tanks. A valved bypass line around the orifices was considered, but, unfortunately, space was limited. An alternative was selected incorporating JUPITER missile prevalves modified with a hole drilled in the butterfly to obtain the necessary orifice diameter when the valve was in the closed position. Opening the valves allowed unrestricted flow of the pressurant gas during fill.

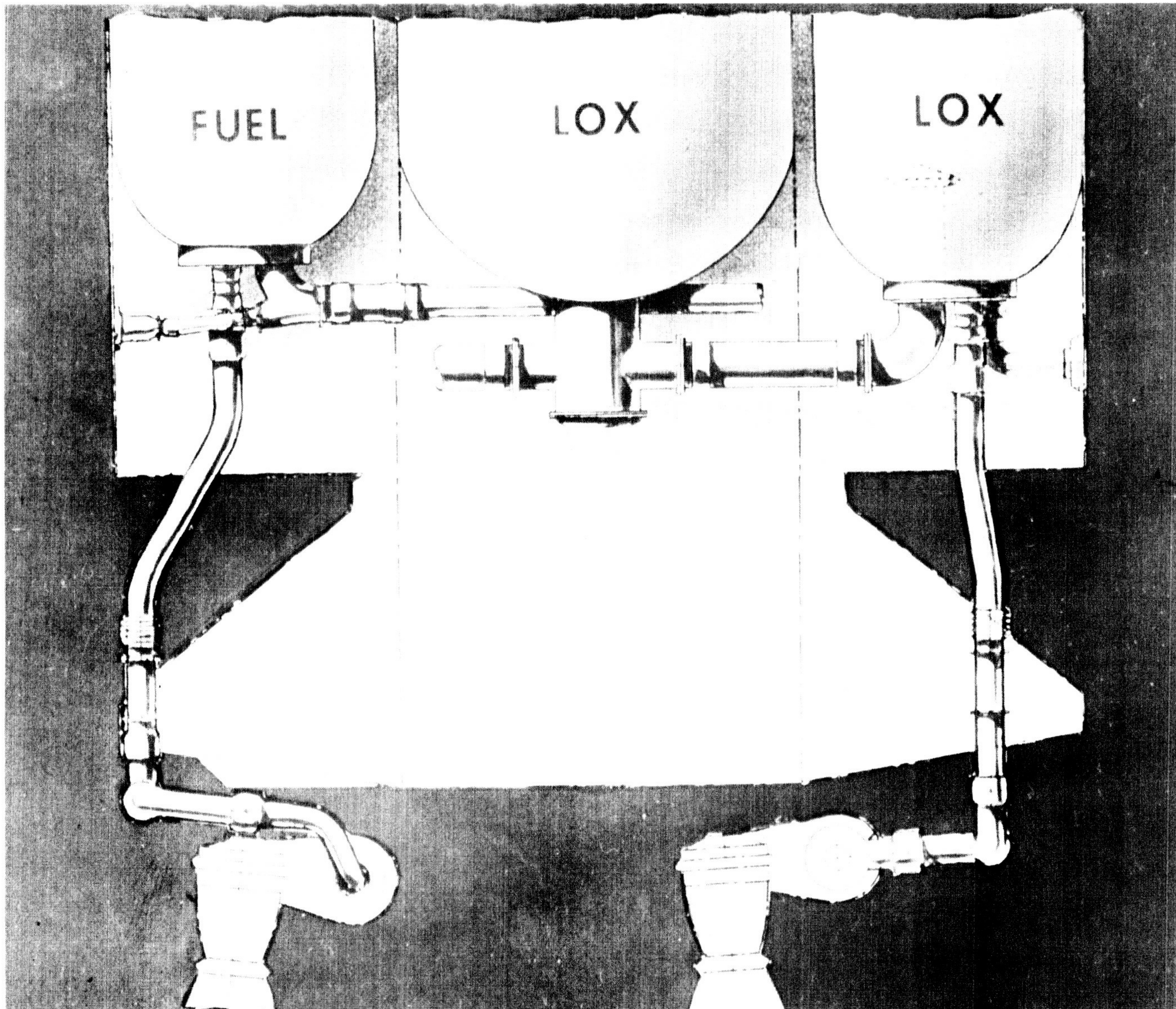

FIGURE 1. SATURN I BLOCK I PROPELLANT INTERCHANGE CONFIGURATION

Following these design modifications, a series of tests were initiated to demonstrate terminal draining of the SATURN I booster. For the first ninety seconds of the first static test in this series all went well. Then suddenly, the pressure in number Three LOX tank dropped from 70 psi to 25 psi within 3 seconds causing the test to be terminated. At this time the standpipes had not been installed in the test vehicle. It was assumed that their absence was the cause of the pressure collapse; therefore, prior to the next test, the standpipes were installed. Again, the same thing happened except the time pressure collapse was seven seconds earlier. Analysis showed that in both cases, the pressure collapse occurred when the liquid level was about 60 inches above the exit of the LOX transfer line. It was decided that the pressure collapse phenomena was the result of a surface disturbance caused by the upward flow of the LOX from the center tank. The surface disturbance caused a rapid increase in the heat transfer between the pressurant gas and the liquid. A rapid pressure drop followed due to the gas temperature being reduced to saturation. This increased the transfer flow to the point where it finally produced a fifty-foot geyser within the tank. A solution was to install a baffle over the end of the standpipes to divert the flow away from the liquid surface. In six days such a device was designed, fabricated and installed on the test vehicle. A subsequent terminal draining test was successful, but during the final six seconds, the tank pressure dropped approximately 5 psi. This final six seconds was the interval in which the liquid level was below the baffles (which, incidentally, have been named "Chinese Hats"). Continuing optimization and scale model testing of the baffle design, based on Russian data, resulted in a configuration which yielded almost unnoticable pressure decay during this interval. Peculiarly enough, this detailed set of test data for the improved version of the "Chinese Hat" was developed by the Russians for optimizing ventilator exhaust ducts on railway cars. The data proved to be quite valid on the highly successful flights of the SATURN vehicle.

SECTION II. PROPELLANT FEED SYSTEM DEVELOPMENT

Some of the earliest design problems of SATURN development stemmed from the selection of the H-1 engine, the first of the large rocket engines carrying the turbopump assembly piggy back. Such a design requires the gimballing motions and flexing forces to be absorbed by the low pressure feed system rather than the high pressure feed system. Although the piggy back arrangement greatly simplified the engine development problems, the design of the low pressure feed system was severely complicated. These lines must absorb engine ignition and cutoff motions to allow for tolerance buildups of installation misalignments, and to permit an engine gimbal pattern of 7 degrees square or 9.8 degrees across the corners.

Based on experience from past programs, preliminary designs considered single and multiple braid-restrained bellows in a wrap-around arrangement. Single bellows were eliminated because the location of the pump inlets required excessive motions. The braid restrained bellows arrangement was rejected because its extreme stiffness when pressurized caused intolerable pump loading. An acceptable design that met allowable pump load requirements utilized gimbal joints that were proven components in the aircraft industry for less severe applications. The incorporation of two gimbal joints 90 degrees apart in the horizontal plane of gimbal and a third joint in the vertical plane to eliminate torque, allowed for gimbal motion of the engine with a minimum motion of the lines.

The state-of-the-art at this time classified the SATURN booster as almost impossibly complex. To assure maximum safety during the development of this stage, it was mandatory that prevalves be incorporated into each of the 16 low pressure propellant feed lines. The only readily available valve that most nearly met the SATURN requirements was the JUPITER prevalve. Although this valve was far from optimum for this application, it was requalified for the more stringent conditions and incorporated into the design. In more than 50 static cluster firings, only once was this valve needed, but this one instance made the cost of including this valve in the design well worth the effort.

The introduction of the SATURN Block II design will bring this stage to a much higher level of refinement, flexibility and performance. The individual engines are uprated from 165,000 pounds to the original design goal of 188,000 pounds. Also, the engine gimbal requirements are increased from 7 degrees square pattern to a 10 degree square pattern. The design of the wrap-around suction line was further improved to permit this increase in the gimbal capability so redesign will not be required if winged payloads are incorporated into the SATURN mission program. At the same time, the propellant feed system for the Block II vehicles will be modified for best routing and minimum pressure drop. As shown in FIG 2, sumps were added to the tank bottoms to minimize propellant residuals, and a low pressure drop ball-type prevalve was incorporated for high reliability and minimum weight. In every case the basic design principles of the Block I vehicles were maintained in the Block II configuration.

Pressure volume compensating joints were considered in the low pressure feed system for the Block II vehicles as a replacement for the wrap-around lines, but they were rejected because the schedule prohibited the necessary vehicle modifications and large number of studies that were required. Subsequent to the freezing of the design on the Block II vehicles, eight-inch pressure volume compensating joints have been built and are being tested for application to the SATURN IB vehicle. Studies are underway to determine the feasibility of using this component on the Chrysler built production type vehicles. Incorporation of pressure volume compensating joints would reduce assembly problems, eliminate many areas of critical clearance, and further improve the design of the low pressure feed system. The desirability of this improvement is obvious when the complex line arrangements in FIG 3 and FIG 4 are considered.

The S-IC, the first stage of the SATURN V vehicle (sometimes called the Advanced SATURN) utilizes five 1.5 million pound thrust F-1 engines to produce a total thrust of 7 1/2 million pounds. The F-1 engine has two fuel inlets of 12 inches diameter and one oxidizer inlet 17 inches in diameter. The F-1 engine gimbal requirements for the S-IC vehicle were established as a six degree square pattern. Propellant feed ducting must allow for installation tolerances, support deflections, and a six degree gimbal pattern that includes over-travel, snubbing, thrust vector adjustment, and control. Various low pressure feed system ducting designs were considered for the preliminary versions of the S-IC before the present design was finally selected.

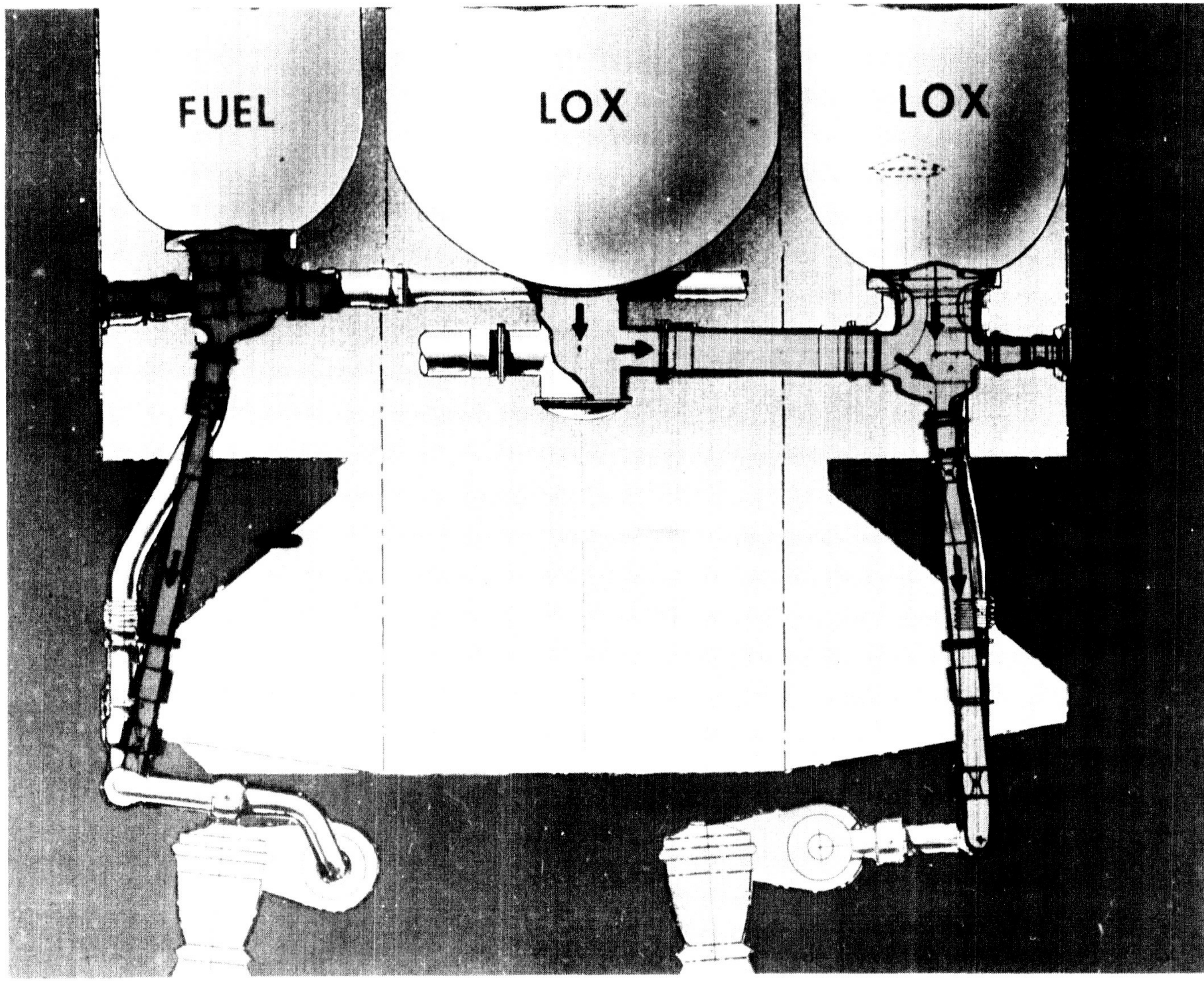

FIGURE 2. SATURN I BLOCK II PROPELLANT INTERCHANGE CONFIGURATION

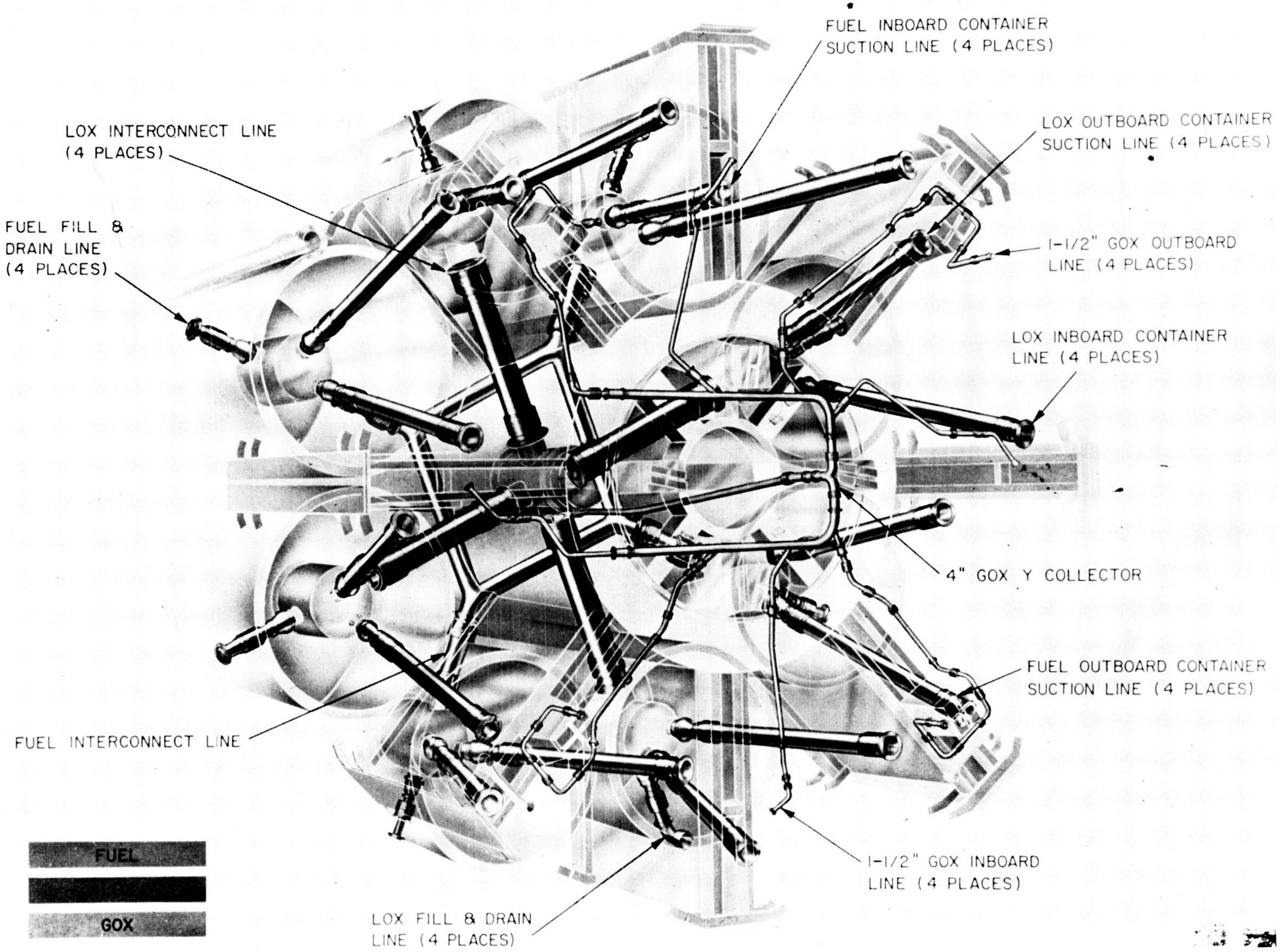

FIGURE 3. SATURN BLOCK II LOW PRESSURE FEED SYSTEM

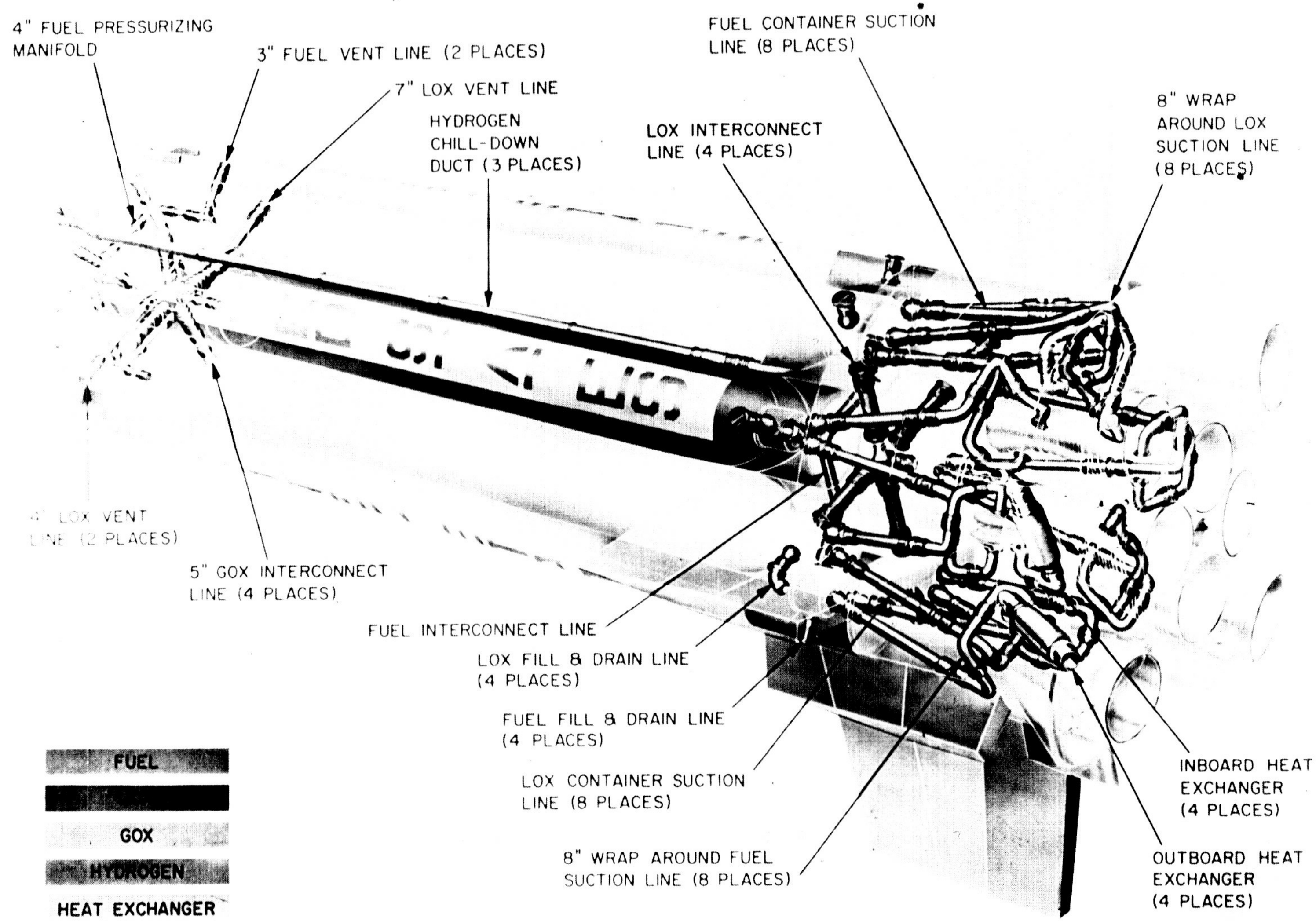

FIGURE 4. SATURN BLOCK II PROPELLANT FEED SYSTEM

The LOX feed lines are routed from the bottom of the LOX tank, through tunnels in the fuel tank, to the engines. The LOX suction lines are 20 inches in diameter with adaptors for a 17 inch prevalve and a 17 inch pressure volume compensating duct. The fuel propellant feed system consists of a 12 inch fuel suction line, 12 inch prevalves, and a 12 inch pressure volume compensating duct.

The incorporation of wrap-around lines in the S-IC was never seriously considered because the requirement for three suction lines for each engine does not permit such a design. Since the S-IC design came later than the SATURN Block II, serious consideration was given to pressure volume compensating ducts, one version of which is shown in FIG 5. The situation, insofar as development time was concerned, was considerably improved by the earlier studies that were done for the Block II configuration. Although less than one year elapsed between the time that the design studies were performed for the SATURN Block II and the time similar studies were performed for the S-IC, the state-of-the-art had advanced sufficiently to assure that the pressure volume compensating ducts would be a reliable design in addition to being more simple and straight-forward, as shown in FIG 6 and 7. These components are presently being developed--the prevalves jointly by AiResearch and Whittaker and the ducting by Arrowhead Products, Company.

SECTION III. PROPELLANT TANK LOCATION FOR S-IC STAGE

Propellant tanks for the S-IC stage are of conventional design with the LOX tank forward and the fuel tank aft. Although this type of configuration is without the problems of clustered tanks, the design analyses required are quite complex. It is difficult to prove by analytical means that the selection of an alternative propellant tank arrangement would be better because of the large number of variables that must be considered.

As a general rule, the larger the vehicle, the less likely it is that there will be a single variable which would unconditionally govern the selection. It is also probable that for vehicles that may be doubled in size, the choice might be determined by an entirely different set of variables. It might be said for a small single stage vehicle, the more dense propellant should be near the center of the vehicle, but for a particular stage of a multi-stage vehicle, the effect of placing the more dense propellant near the geometrical center of the vehicle is of considerably smaller significance.

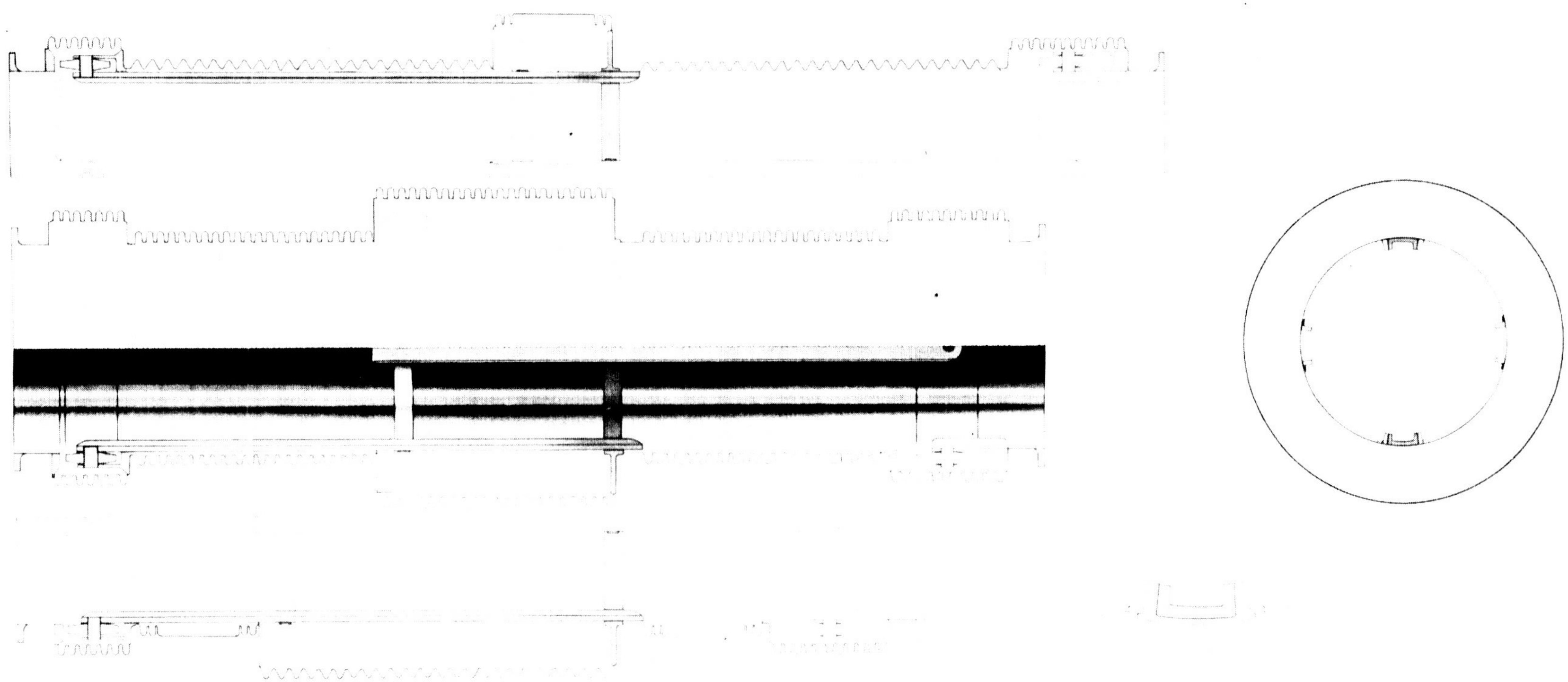

FIGURE 5. SATURN V PRESSURE BALANCED JOINT

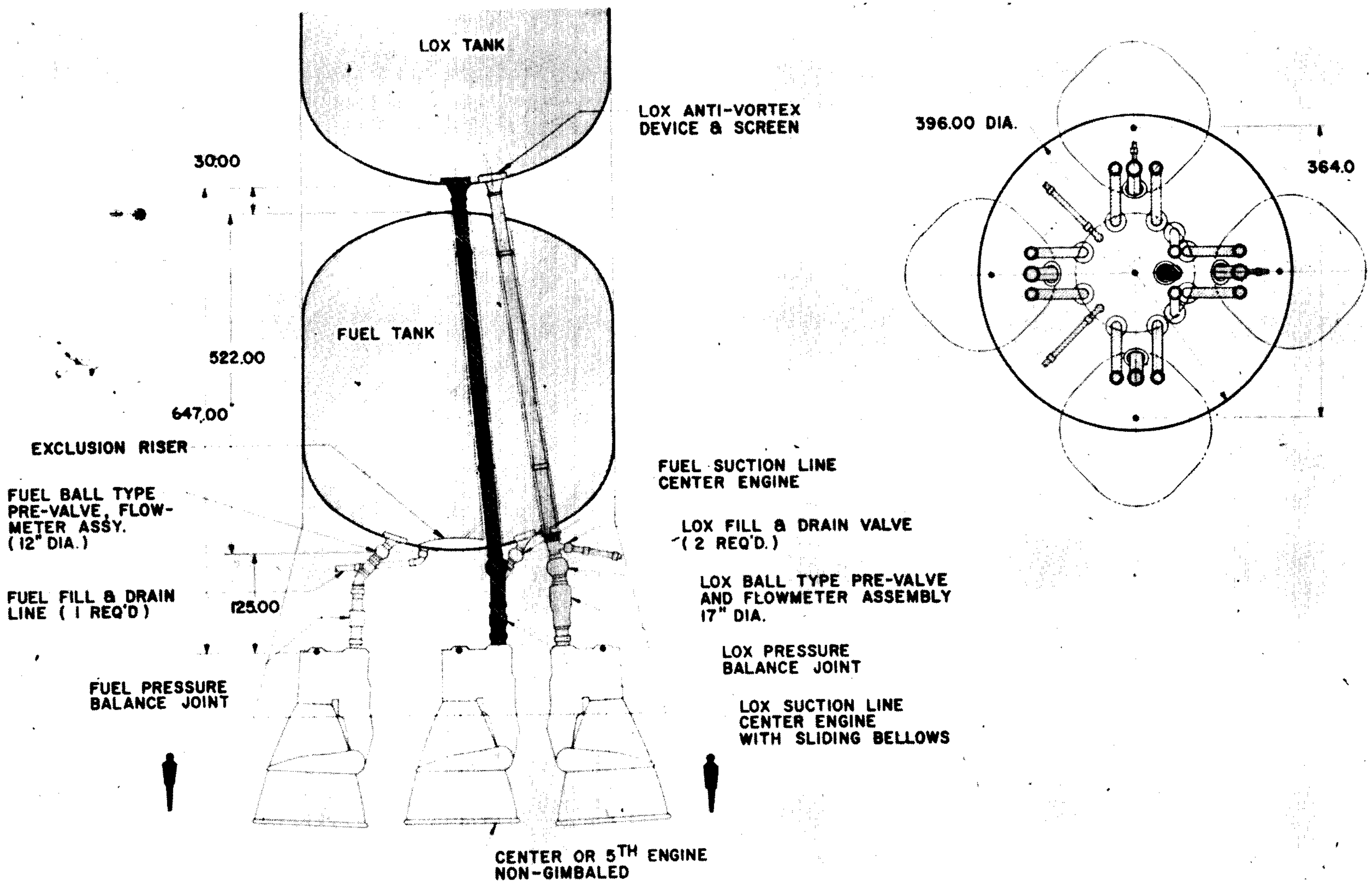

FIGURE 6. SATURN V PROPELLANT FEED, FILL, AND DRAIN SYSTEMS

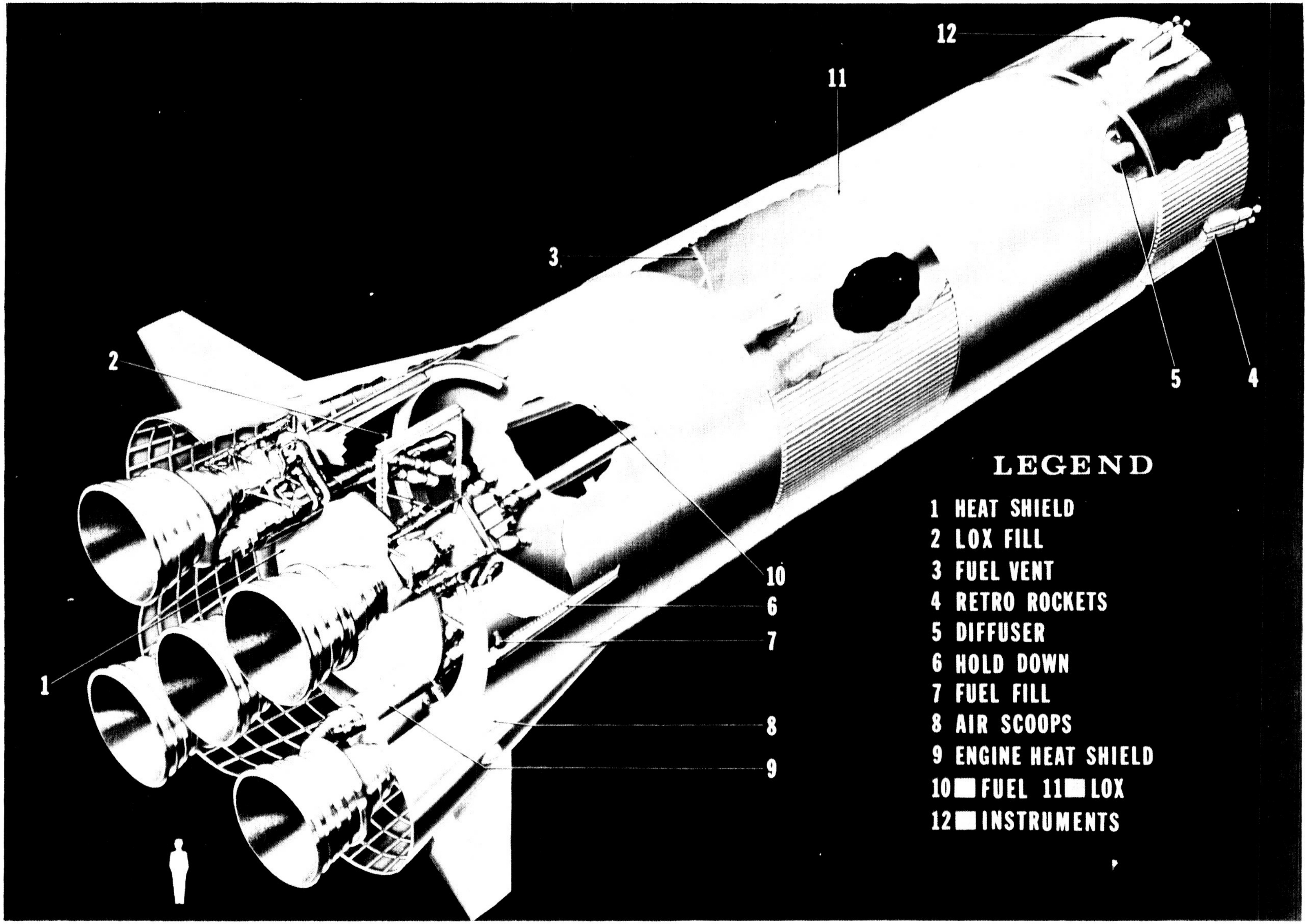

FIGURE 7. SATURN V BOOSTER

The major factors affecting the selection include the total system weight, suction line relief and routing, liquid residuals, pressurization requirements, stability, sloshing effects, and a more obscure area best described as mutual influences of the engine and the tankage upon each other. It is also necessary to consider the fabrication problems of the larger vehicles. One of the more formidable difficulties is finding a building large enough in which to build, assemble, and calibrate such a stage.

A typical example of weight comparison is illustrated in FIG 8 which presents the total system weights for the S-IC stage for the "LOX Tank Forward" versus the "LOX Tank Aft." The weight advantage shown for the LOX Tank Forward is quite significant since it saves about 3,000 pounds of payload. Separated bulkheads were assumed in this comparison, but even this choice is not a simple one.

In the final analysis, the decision to use separated tanks, rather than a common intermediate header, was dictated by fabrication requirements. Additional pressurization was required to prevent the reversal of the tension header during flight, and facilities large enough to house the total tankage during water calibration were non-existent. The selected design permits the assembly of the two tanks in the horizontal position after each tank has been individually volume-calibrated in the upright position. This configuration also permits removal of the suction line without tank entry.

Short low-pressure feed lines are desirable to reduce the weight of the lines and liquid residuals. Also, thrust and specific impulse can be expected to drop during the latter portions of flight with the "LOX Tank Aft," whereas with the "LOX Tank Forward," an increase in thrust is realized without much change in specific impulse. This effect is caused by the sensitivity of the F-1 engine to variations in LOX and fuel pump inlet pressures that are in turn highly dependent upon the length of the lines and the acceleration profile of the stage during flight. The sloshing effect was also found to be less severe with the LOX Tank Forward because the sloshing mass in the LOX tank is in a critical zone for a shorter interval, approximately 25 seconds, in comparison to 100 seconds for the LOX Tank Aft.

The critical slosh zone is the area between the center of vehicle instantaneous rotation and the vehicle center of gravity. When the

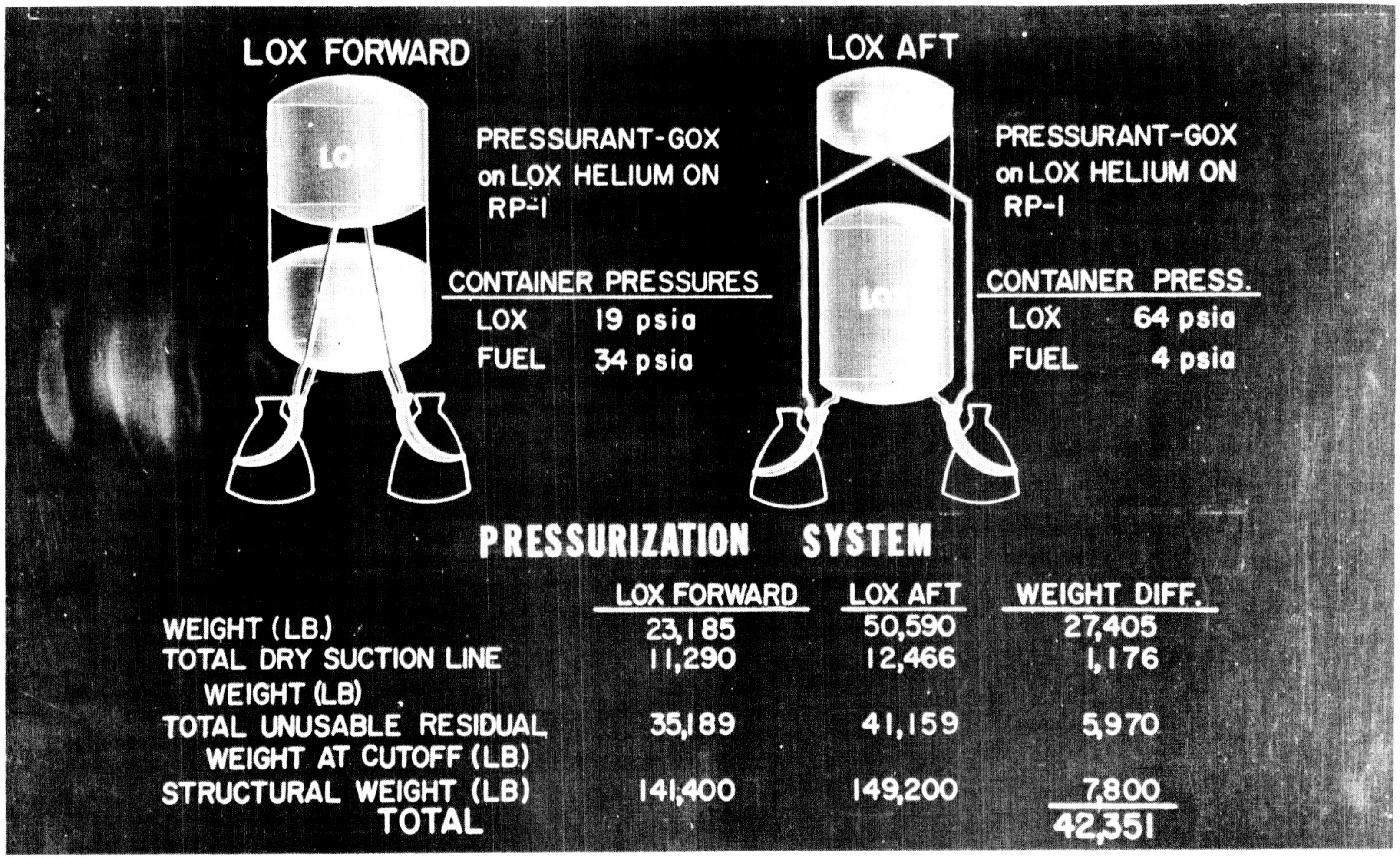

FIGURE 8. PROPELLANT TANK LOCATION LOX FORWARD VERSUS LOX AFT, SATURN V, S-IC STAGE

liquid level is in this area, the highest amplitude of slosh can be expected. The relative position of the propellant tanks was found to have very little influence on the stability of the vehicle during flight. However, because the center of gravity is shifted with LOX tank forward, and because there is less motion of the center of gravity during flight, the stability of the LOX tank forward configuration was judged to be slightly better than that of the LOX tank aft. From these conclusions the LOX tank was placed forward.

SECTION IV. PROPELLANT CONDITIONING FOR ENGINE START

To achieve a satisfactory start, the Pratt and Whitney RL10A-3 engine, used on the SATURN S-IV stage, requires pre-chilling of the turbopump assembly to temperatures approaching the propellant temperatures. At the present time, this requirement is satisfied by dumping raw propellants through the engine in the pre-start phase. This type of operation, considered to be extremely hazardous, was thoroughly investigated. It was found that these combustibles might accumulate and ignite; (1) in the S-IV/S-I interstage (especially during separation when the LH_2 dumping from the disconnected vent stack could freeze because of expansion and form a high explosive yield) solid hydrogen, solid oxygen mixture; (2) between the cluster tanks of the booster stage; (3) in the boundary layer adjacent to the first stage skin; or be drawn into the first stage engine compartment and ignited. Since the SATURN was designed primarily for manned payloads, these possible sources of destruction to the vehicle could not go uncorrected.

Modifications of the S-IV propulsion system were considered, assuming it was not necessary to dump propellants overboard to chill the engine. These schemes were:

a. Engine pre-chill prior to lift-off

b. Propellant recirculation with pumps

c. Internal pump insulation

d. External pump cooling jacket

After feasibility testing none were adopted since improvements would be required and because the systems were complex.

At this point, it was concluded that the fastest, most reliable solution of this problem would be the installation of three twelve-inch diameter ducts running the length of the S-I stage to allow the combustible hydrogen gases to be dumped overboard below the clustered containers. Although these ducts were quite heavy, the net increase in the weight of the booster was negligible. Increased understanding of the structural problems of clustered tankage led to a considerable reduction in the weight of the stage itself.

The LOX used for chilling down the RL10A-3 is vented through the thrust chamber rather than a separate manifold, as in the case of the hydrogen side. At the altitude of separation, pressures in the interstage area are below the triple-point of oxygen, and it is certain that liquid exhausted through the nozzle will solidify. Because there are always ignition sources present in any rocket-propelled device, the presence of solid oxygen almost assures that ignition will occur if even the smallest amount of fuel is available. Late in the development phases of the Block II design it was specified that there would be no accumulation of solid oxygen in the interstage area.

The engine design did not lend itself to the adaptation of vent ducts or circulation of LOX as it does in the case of hydrogen. It was concluded that it would not be possible to direct the flow of the vented oxygen to some specific area, as it was necessary to compromise to the point that solid oxygen would be eliminated by gasification of the cooldown LOX. To achieve this, a GN_2 system was developed to gasify the LOX while it was still in the engine nozzle. In order to optimize the amount of GN_2 that would be necessary to perform this function, an extensive test program was conducted. The tests demonstrated that the GN_2 purge could preclude solid LOX (SOX) from forming.

In the original design of the SATURN Block II it was assumed that recovery of the booster might become part of the design. For this reason an excess amount of GN_2 is carried onboard to insure that the pressure in the fuel tanks is above ambient sea level conditions. Only about sixty percent of this total amount is required for fuel tank pressurization during the powered portion of the booster flight. The remainder of this warm nitrogen is used in what has since come to be known as the LOX/SOX disposal system. Unfortunately, the LOX chilldown flow was determined to be somewhat greater than originally anticipated so it was necessary to add additional nitrogen onboard. The locations of the nitrogen sphere, pressurization lines, and hydrogen chilldown ducts are shown in FIG 9.

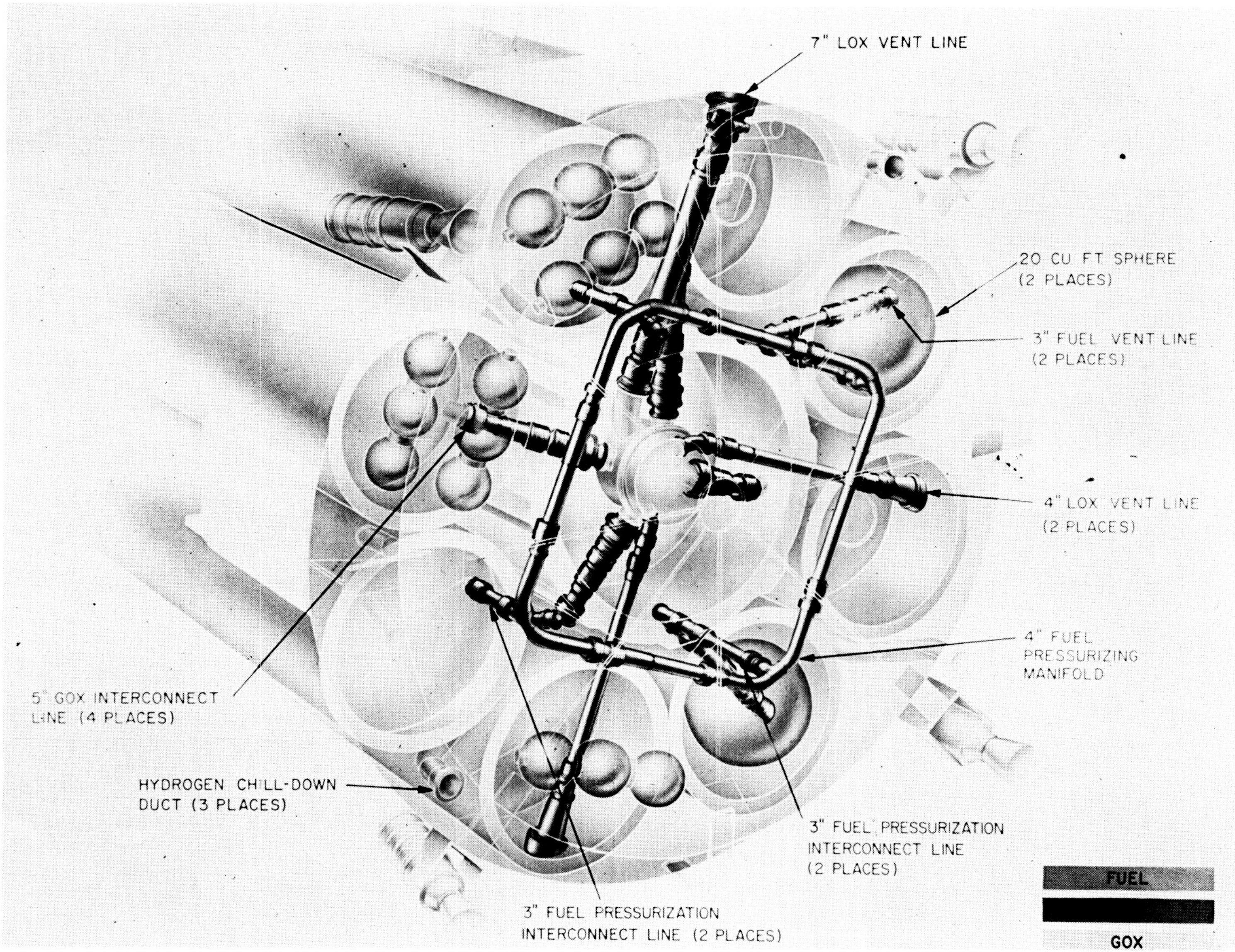

FIGURE 9. SATURN BLOCK II PRESSURE AND DISTRIBUTION SYSTEM

Because of space limitations, it was possible to carry oniy enough nitrogen to last for ten seconds. This limitation on turbopump pre-chilling time necessitated the establishment of a maximum value of the LOX pump inlet temperature to 168°R to permit an adequate engine start. To assure this temperature is not exceeded, cold helium is injected into the LOX suction lines prior to lift-off. Preliminary tests established the cold helium flowrate necessary to produce LOX temperatures at between 135 and 140 degrees R at lift-off, so the maximum temperature of 168°R at booster burnout may not be exceeded.

Again, because of the one-year difference in time frame between Block II and the advanced SATURN, it was possible to solve the problems listed above by more refined methods than were used on the Block II. The main approach for engine chilldown on the advanced SATURN was to incorporate the concept of propellant recirculation. The upper stages of the advanced SATURN use engines with wet pumps (propellant valves below pumps) that begin cooling when the propellants are loaded. Circulation is by means of submerged electrically driven pumps that have a size and weight advantage because of increased electrical conductivity at cryogenic temperatures. These pumps will also ensure that good quality propellants, containing a minimum amount of gas, will be available to the engine pumps at any time. Thus, minimum effort will be required to develop acceptable propellant suction line insulations, and maximum confidence will be afforded in the engine start transients due to controllable and repeatable start parameters of pump body temperatures and propellant quality. Circulation, possibly, can be by natural convection. Indications are that natural convection circulation, promoted by differential heat leaks between parallel lines, is an effective deterrent to geysering or percolating that has been noted in long, vertical cryogenic lines.

SECTION V. BASE HEATING

In addition to radiant heating from engine exhaust gases, common to all boosters, vehicles with clustered engines may experience significant convective heating in the base region, because of backflow of engine exhaust gases caused by jet interaction. A further source of base heating may be combustion of fuel-rich gases entrained in the base region. These gases may come from the engine exhaust, turbine exhaust, or the fuel vent system.

In discussing cluster engine base heating, it is common to define three general flow regimes. These regimes will be described in terms of the heat transfer to a "cold" receiver, or sink. Otherwise, it would be necessary to define the receiver transient temperature that is influenced by the receiver thermal and physical properties as well as the thermal environment. The first regime applies to the condition that is typical just after lift-off, where the jets do not interact with each other. The jets act as ejectors and cause air to be drawn into the base region as shown in FIG 10a. When hydrocarbon-oxygen exhaust products are involved, radiation is by far the dominant mode of heat transfer in this regime. Afterburning of fuel-rich gases in and around the engine exhausts can significantly increase exhaust radiation. Engine exhaust gases are generally fuel-rich because of engine performance considerations. For instance, the SATURN S-I stage engines operate at an oxygen-to-fuel ratio of approximately 2.4 as opposed to the stoichiometric ratio of 3.4. This presents an opportunity for afterburning of the jet as it mixes with the ambient air. Fuel-rich turbine exhaust gases are another possible source of afterburning. SATURN S-I turbines operate at an oxygen-to-fuel ratio of about 0.3. Although some backflow of exhaust gases may occur because of jet and free stream interaction, convective heating is generally negligible compared with radiation in the first regime (a possible exception might be when hydrogen-oxygen propellants are used and radiant heating is also low). As altitude is increased, the jets expand because of the lower ambient pressure. Finally, an altitude is reached where the jets interact with each other and the external stream to cause a backflow of exhaust gases into the base region as shown in FIG 10b. This marks the beginning of the second regime and is characterized by significant convective heating of the base. As altitude is further increased, the jets continue to expand, decreasing the flow area through which the backflow gases escape from the base region. Finally, the area is decreased to such an extent that sonic velocity is attained at the location of minimum flow area that generally exists between adjacent nozzles. After the beginning of choked flow characterizing the third regime, the base pressure is no longer affected by changes in the ambient pressure, and convective heating is at a near-maximum value and remains essentially constant as altitude is increased. To summarize, the three regimes are characterized by (a) non-interacting jets, related to low altitude operation where the dominant mode of heat transfer is radiation, (b) interacting jets, where convection is significant and increases with altitude, and (c) choked flow, where convective heating reaches a near constant maximum value.

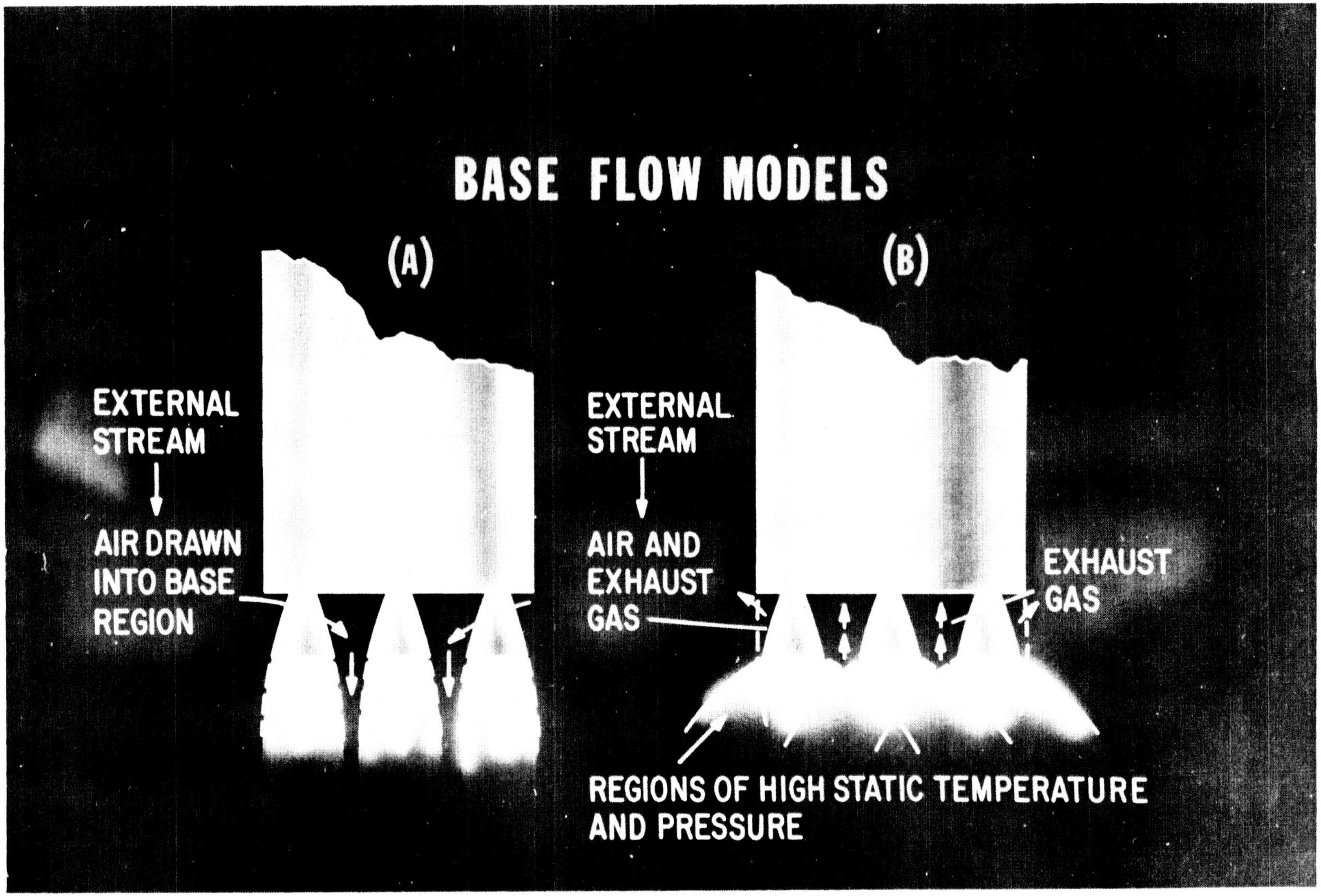

FIGURE 10. BASE FLOW MODELS

In general, radiation to the vehicle base from the jets decreases with increased altitude. This is caused by the increased overall radiation view factor of a point on the base to the expanded jets that tends to increase radiation and is more than offset by the decrease in radiation caused by the reduced-jet static temperatures. Further, the region of afterburning of fuel-rich engine exhaust gases moves further downstream in the jet, and afterburning of fuel-rich gases from the engines plus the other sources mentioned, decreases with altitude as the amount of atmospheric oxygen available for combustion decreases. Finally, an altitude is reached where afterburning ceases.

No analytical model is available to rigorously define jet radiation characteristics. Pressure and temperature vary throughout the jet and change with increasing altitude; generally the jet is neither completely opaque nor does it radiate as a true grey body. The situation is further complicated by the changing degree and location of afterburning. This makes analytical prediction of radiant heat transfer very difficult and encourages the use of experimental and empirical prediction techniques. FIG 11 indicates the decrease in radiant heat flux with altitude at a point on the SATURN S-I heat shield, as determined from inflight measurements.

As stated before, convective heating is small prior to the altitude at which backflow begins, and increases with altitude to a relatively constant value at the beginning of choked flow.

FIG 12 shows the engine arrangements for the SATURN I and V vehicles. Since all SATURN upper stages use LH_2 - LOX propellants, radiant heat transfer occurs in the discrete wave length bands of radiation from water vapor. This, and the effect of high altitude operation on radiation, makes upper-stage radiant heating much less severe than that of the first stages that use LOX, RP-1 propellants. First-stage exhaust jets radiate over a wide wave length spectrum because of the dominating effect of the solid carbon particles in the exhaust. Upper-stage base heating is, then, almost altogether caused by convection. Since the upper stage engines operate only at high altitude, convective heating from exhaust gas backflow is not aggravated by combustion of fuel-rich gases in the base region because of the absence of atmospheric oxygen necessary to support afterburning. Table I shows the predicted maximum heating rates for SATURN upper stages with clustered engines. For comparative purposes, heating rates are based on cold wall conductors. Comparison of the values indicates that upper stage base heating is relatively mild. In contrast with the upper stages, SATURN I and SATURN IC base heating is largely caused by radiation.

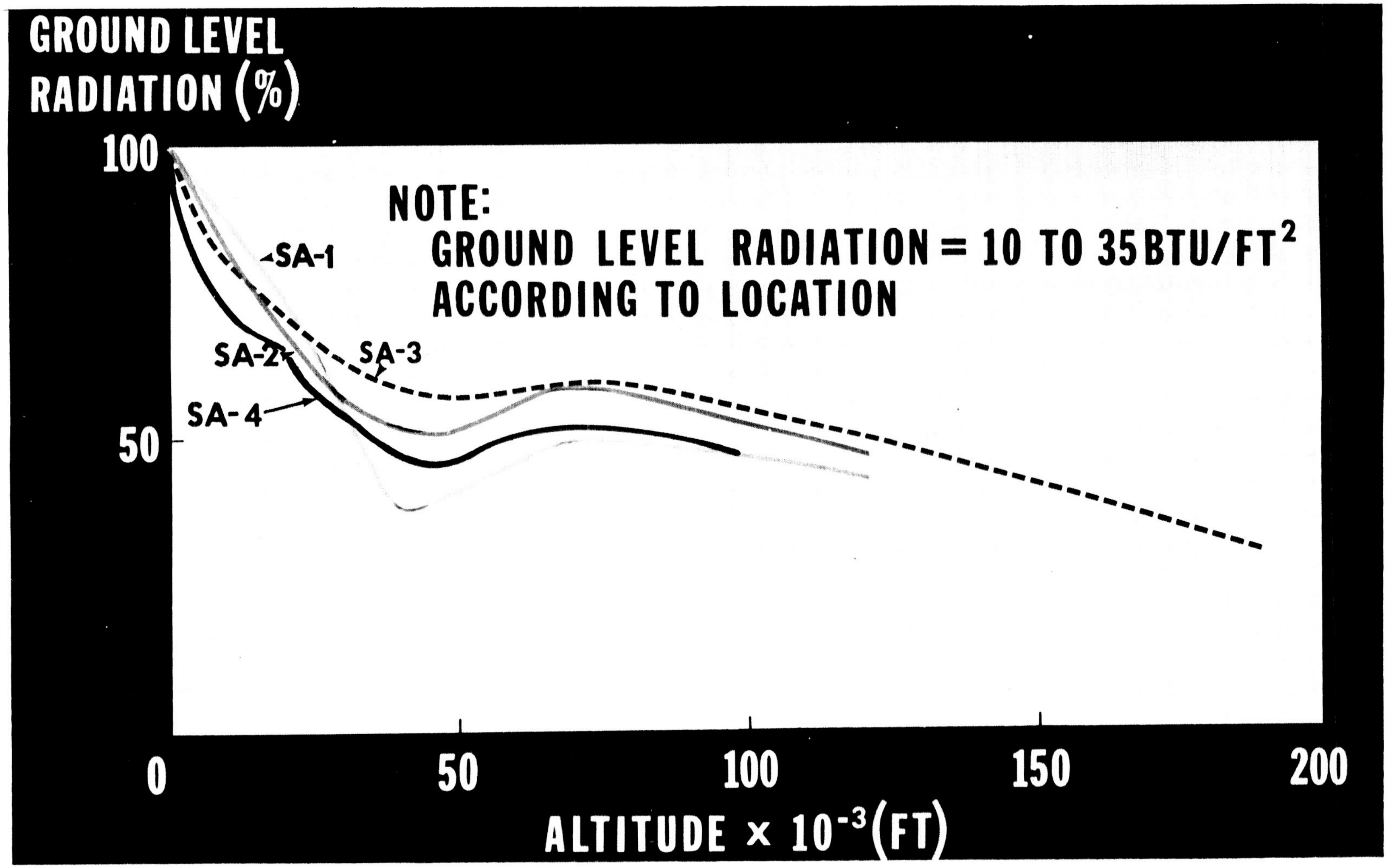

FIGURE 11. RADIATION VARIATION WITH ALTITUDE, S-I STAGE BASE HEAT SHIELD

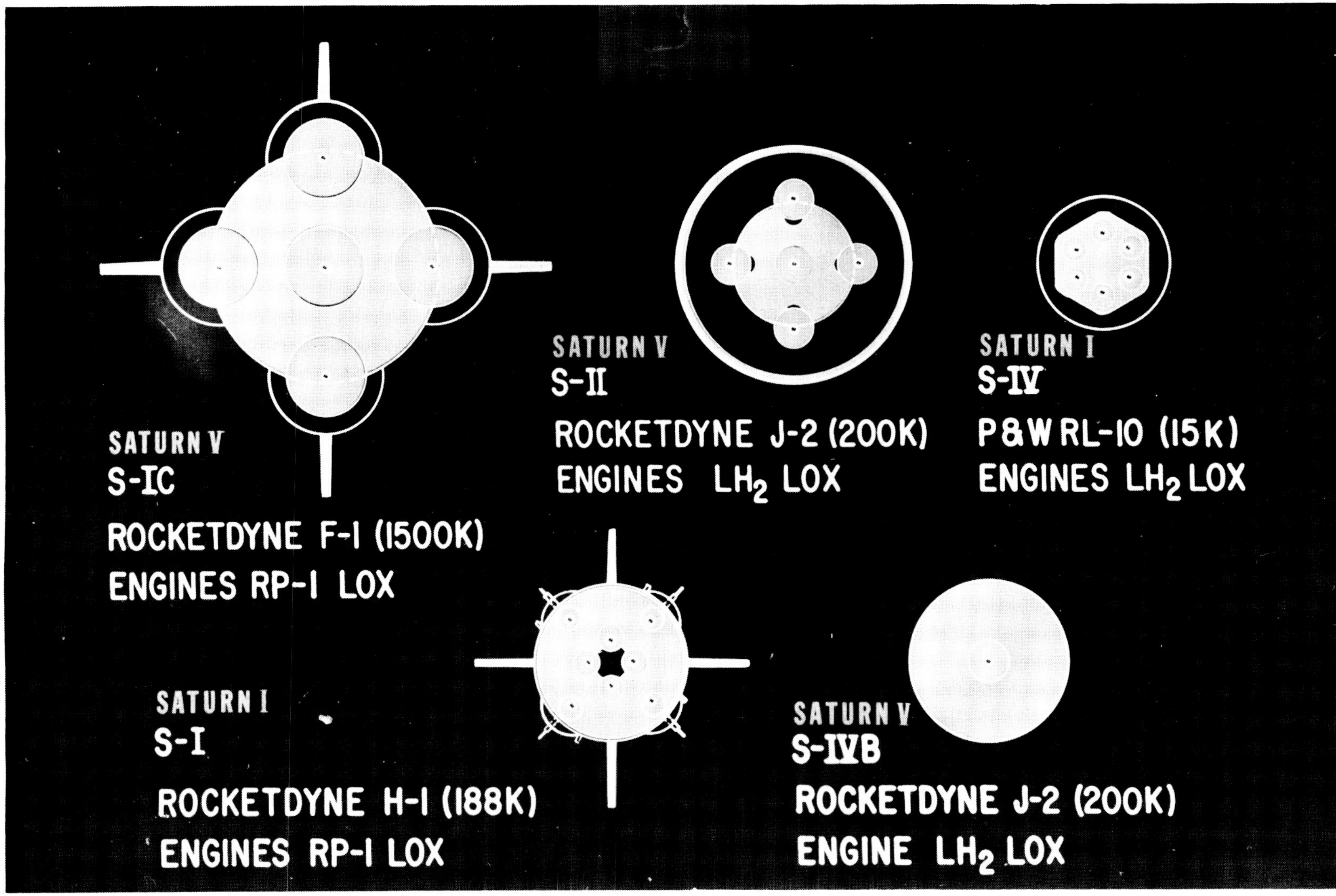

FIGURE 12. SATURN ENGINE CONFIGURATIONS

TABLE I

Maximum Heat Flux to Base Heat Shield
(Based on $100^{\circ}F$ Wall Temperature)

BTU/ft^2 sec

S-I	45
S-IC	35
S-IV	4
S-II	4

As an example of clustered engine stage base thermal protection, the S-I heat shield design will be described.

Protection of the S-I stage base is accomplished by the use of a rigid heat shield between the severe engine exhaust thermal environment and the base structure and vulnerable components. The heat shield consists of high temperature insulation backed up by a metallic support structure. Flexible curtains are used to preclude hot gases and radiation from the exhausts entering the heat shield cut-outs around the engines. These cut-outs are required to prevent engine damage due to interference with the shield during load transients and to allow gimballing of the outboard engines for vehicle control.

FIG 13 shows the S-I heat shield and flexible curtain design. The material designated, M-31, was developed by the Engineering Materials Branch of the Propulsion and Vehicle Engineering Division. M-31 is a trowelable mixture of potassium titanite and silica with 10% asbestos fibers. This material is attractive for heat shield insulation because of its easy application and high-temperature operating capabilities. The density of M-31 is 31 lb/ft^3. CT-301, a phenolic and asbestos ablating material, is used in the flame shield area where relatively severe heating exists because of the close inboard engine spacing. Use of this material is limited to regions of severe heating because of its relatively high density (112 lb/ft^3) and cost of fabrication. The flexible curtains use fiberglass insulated with silicone rubber as the primary curtain-load-carrying material. A reflective coating is applied at the external surface to minimize the absorption of radiation.

SATURN flights to date have been used to prove the adequacy of and improve the S-I base thermal protection. Flexible curtains and flame shields on SA-1, 2, and 3 were designed to be used on the later Block II and operational vehicles. The heat shield used on these Block I vehicles was an interim design heavier than the M-31 insulation Block II design. However, one M-31 insulation panel was installed and flown on SA-3 and proved to be completely satisfactory.

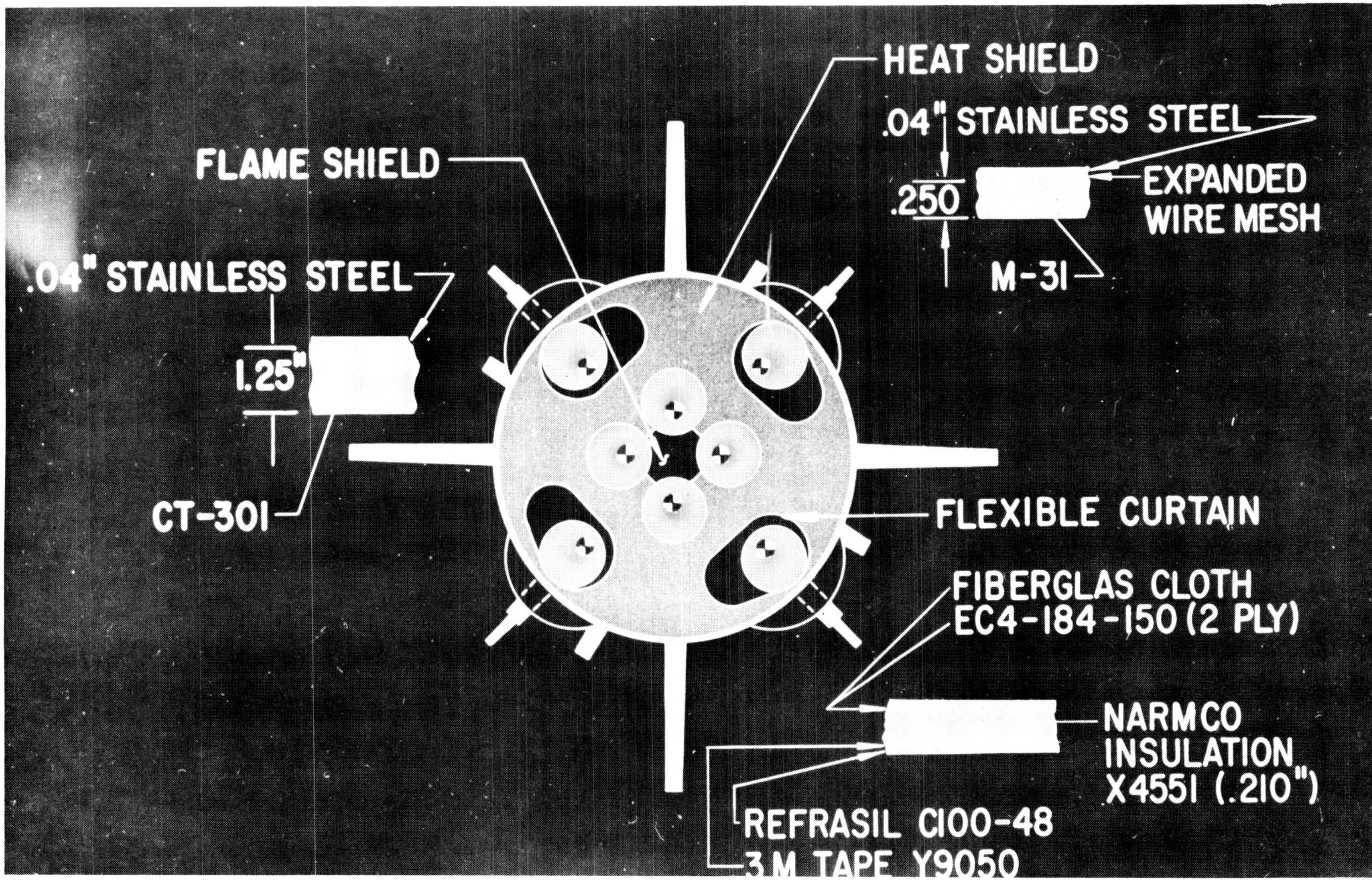

FIGURE 13. S-I BASE THERMAL PROTECTION

NASA TM X-53075 APPROVAL August 12, 1963

PROPULSION DEVELOPMENT PROBLEMS ASSOCIATED WITH LARGE LIQUID ROCKETS

by

A. A. McCool

G. H. McKay, Jr.

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report has been determined to be Unclassified.

George H McKay Jr

G. H. McKay, Jr.
Deputy Chief, Propulsion Evaluation Section

A A McCool

A. A. McCool
Deputy Chief, Propulsion and Mechanics Branch

H G Paul

H. G. Paul
Chief, Propulsion and Mechanics Branch

W A Mrazek

W. A. Mrazek
Director, Propulsion and Vehicle Engineering Division

DISTRIBUTION

DIR
Dr. von Braun

DEP-T
Dr. Rees

R-FP
Mr. Koelle

R-RP-DIR
Dr. Stuhlinger

I-MICH-MGR (2)
Dr. Constan

R-DIR
Mr. Weidner

R-AERO-DIR
Dr. Geissler

R-AERO-P (2)
Mr. Jean

R-AERO-P
Mr. Teague

R-AERO-D
Mr. Horn

R-AERO-F
Dr. Speer

R-AERO-G
Dr. Hoelker

R-ASTR-DIR
Dr. Haeussermann

R-ASTR-TJ
Mr. Noel

R-ASTR-E
Mr. Fichtner

R-ASTR-F
Mr. Blackstone

R-ASTR-G
Mr. Mandel

R-ASTR-I (2)
Mr. Hoberg

R-ASTR-M
Mr. Boehm

R-ASTR-N
Mr. Moore

R-ASTR-NG
Mr. McMahen

K-DIR
Dr. Debus

K-P
Col. Petrone

K-V
Dr. Gruene

D-D
Mr. Poppel

K-DP
Mr. Wasileski

K-EP
Mr. Collins

K-AD
Mr. Zeiler